UNIVERSITI PUTRA MALAYSIA

TORQUE MODE-BASED FLYWHEEL SYSTEM FOR SMALL SATELLITE ATTITUDE CONTROL

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Thesis Submitted to the School of Graduate Studies, Universiti Putra Malaysia, in Fulfilment of the Requirements for the Degree of Master of Science

May 2005

Dedicated to my family

Abstract of thesis presented to the Senate of Universiti Putra Malaysia in partial fulfilment of the requirements for the degree of Master of Science

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May 2005

Chairman: Renuganth Varatharajoo, PhD

Faculty : Engineering

One of the most important problems in satellite design is that of attitude control. An architecture of attitude control for small satellite using flywheel system is proposed. This flywheel system can combinedly perform both the energy and attitude control task that are the crucial area for all types of satellites. Combining the conventional energy and attitude control system is a feasible solution for small satellites to improve the space missions. In this combined energy and attitude control system (CEACS) a double rotating flywheel in the pitch axis is used to replace the conventional battery for energy storage as well as to control the attitude of an earth oriented satellite. Each flywheel is to be controlled in the torque mode. The energy and attitude inputs for the flywheels' control architecture are also in the torque mode. All related mathematical representations along with the relevant transfer functions are developed. The required numerical calculations are performed using Matlab™ for studying the system performances. The goals of this work are to determine the CEACS attitude

performance in the torque mode with respect to the ideal and non-ideal test cases for chosen reference missions, i.e., Nanosatellite (10 kg), Microsatellite (50 kg) and Enhanced Microsatellite (100 kg). The test results concerning to the entire satellite test cases are satisfactory and the ideal/non-ideal CEACS attitude performances coincide with the reference mission requirements. The simulation results show that the torque mode CEACS is able to achieve a good pointing error for small satellites.

Abstrak tesis yang dikemukakan kepada Senat Universiti Putra Malaysia sebagai memenuhi keperluan untuk ijazah Master Sains

SISTEM RODA UNTUK KAWALAN ATITUD SATELIT KECIL DALAM MOD KILASAN

Oleh

IBRAHIM MUSTAFA MEHEDI

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Salah satu daripada masalah yang penting di dalam rekabentuk satelit adalah kawalan atitud. Rekabentuk berkenaan dengan kawalan atitud untuk kegunaan satelit yang kecil menggunakan sistem roda tenaga dicadangkan. Sistem roda tenaga ini dapat menggabungkan sistem tenaga dan sistem kawalan atitud dalam perlaksaan tugas yang mana keduanya adalah bahagian yang paling kritikal untuk semua jenis satelit. Penggabungan tenaga dan kelakuan sistem kawalan konvensional merupakan satu kaedah penyelesaian untuk penggunaan satelit kecil bagi meningkatkan misi angkasa. Di dalam penggabungan sistem kawalan tenaga dan atitud (CEACS), sepasang roda tenaga berkembar berputar di paksi mencancang digunakan untuk menggantikan bateri konvensional untuk penyimpanan tenaga. Setiap roda tenaga dikawal di dalam mod kilasan. Input tenaga dan atitud untuk rekabentuk kawalan roda tenaga juga di dalam mod kilasan. Semua persamaan matematik yang berkaitan serta fungsi penghantaran dibangunkan. Pengiraan berangka yang diperlukan dilaksana menggunakan Matlab™ untuk mengkaji prestasi sistem. Objektif untuk pengajian ini adalah untuk menentukan perlaksanaan atitud CEACS di dalam mod kilasan berdasarkan ujian sistem sempurna dan ujian tidak sempurna untuk suatu misi pilihan, seperti satelit nano (10 kg), satelit mikro (50 kg), satelit mikro (100 kg). Hasil pengujian sistem adalah memuaskan dan ujian sistem sempurna/tidak sempurna mematuhi keperluan misi pilihan. Hasil simulasi menunjukkan bahawa CEACS di dalam mod kilasan mampu mencapai kesalahan atitud yang baik untuk satelit bersaiz kecil.

ACKNOWLEDGEMENTS

First of all my thanks and gratitude to Allah for the completion of this project successfully. I would like to express my heart-felt thanks and sincere gratitude to my supervisor Dr. Renuganth Varatharajoo and the members of the supervisory committee, Mohammad Nizam Filipski and Assoc. Prof. Mohd. Ramly Mohd. Ajir for seeing through this research work.

In addition, I would like to extend my sincere appreciation to my family and friends for making this project a success.

I certify that an Examination Committee met on 18th May 2005 to conduct the final examination of Ibrahim Mustafa Mehedi on his Master of Science thesis entitled "Torque Mode-based Flywheel System for Small Satellite Attitude Control" in accordance with Universiti Pertanian Malaysia (Higher Degree) Act 1980 and Universiti Pertanian Malaysia (Higher Degree) Regulations 1981. The Committee recommends that the candidate be awarded the relevant degree. Members of the Examination Committee are as follows:

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DECLARATION

I hereby declare that the thesis is based on my original work except for quotations and citations, which have been duly acknowledged. I also declare that it has not been previously or concurrently submitted for any other degree at UPM or other institutions.

IBRAHIM MUSTAFA MEHEDI

Date: 19/07/ 2005

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LIST OF ABBREVIATIONS

LEO	Low Earth Orbit
GEO	Geosynchronous Equatorial Orbit
EO	Elliptical Orbit
PD	Proportional Derivative
ACS	Attitude Control System
GPS	Global Positioning System
DoD	Depth of Discharge
FES	Flywheel Energy Storage
CARES	Combined Attitude, Reference, and Energy Storage
MEMS	Micro Electro Mechanical Systems
AOC	Attitude and Orbit Control
BCR	Battery Charge Regulator
BDR	Battery Discharge Regulator
RTG	Radio-isotope Thermoelectric Generator
S^3R	Sequential Switching Shunt Regulator

NOMENCLATURE

Ω_w	flywheel output speed
$I_{w,}\ I_s$	flywheel and satellite inertias
T_s	torque exerted on the satellite body
T_D	external disturbance torques
$v \cong T_{cmd}$	proportional attitude torque command
$T_{energy.cmd}$	energy torque command
$\theta_{ref.}, \theta_y$	reference and satellite attitude
$K_{1,} K_{2,} K$	gains
K_P, K_D	proportional and derivative attitude control gains
$\omega_{n,}\ \xi$	attitude loop natural frequency and damping ratio
I_x, I_y, I_z	moment of inertias Roll, Pitch and Yaw axes
$\mu_\oplus$	earth gravitational constant (3.986005×10^{14} m^3/s^2)
Ω_0	orbital frequency (circular orbit, $\Omega_o = \sqrt{\frac{\mu_\oplus}{r_e^3}}$)
Ω_{cmd}	flywheel angular speed command
K_w	proportional speed controller
r_e	satellite orbit radius
$T_{ecl.}$	orbit eclipse period
P_w	power
C_s	solar radiation constant (1358 W/m^2)

CHAPTER 1

INTRODUCTION

1.1 Introduction

A satellite is an object that orbits another object or a planet. Gravity pulls the satellite closer to the primary object it orbits, but the satellite moves perpendicular to that pull, at a high speed so that it continuously avoids colliding with the primary object. Satellites can be either natural or artificial. Artificial satellite is a man-made one that orbits around the earth or the moon or other celestial bodies to carry out some precise operations or missions.

Artificial satellites are utilized for different purposes like distribution of television and audio signals, telephone connections via satellite are done by communication satellites. Navigation satellites are of enormous help for transport companies, especially transportation over water and through air. The US GPS satellites can determine the position of the object with a precision of 1 cm [1]. Navigation satellites are also used for distance measurements for instance between buildings. The task of weather satellites is to observe the earth and the changes in the atmosphere. Various kind of cameras, like infrared and normal are used to observe either the same part of the earth, from a geostationary orbit or more closely from polar orbits to get more detailed pictures. These low orbit weather satellites focus more on the study of the atmosphere than on the current weather. Very similar to weather satellites, military satellites are also used for observing military installations on the earth. Generally, these satellites are equipped with higher resolution cameras, instead of normal communication equipments. Sometimes such satellites have a very different type of orbit like a elliptical orbit in which it takes the satellite as far away from the earth as the moon and as close to the earth that it shortly enters the atmosphere, to get as close as possible to the earth surface without falling back to earth. Probably many more tactics are used, but for obvious reasons, these are confidential. Observing the earth for scientific purpose is also possible by satellites. Making maps with low polar orbit satellites for instance, but also measuring the exact shape of the earth, geological research, etc can all benefit greatly from scientific satellites.

In the early days of space exploration, most space missions were small, primarily because the launch capability was small. As the launchers grew, so did the satellites. However, it is not forgettable that an incredible increase of human knowledge came from those early small satellites. The need to return to smaller missions was therefore initiated by the space community, which was then being consolidated by the reduction in space budgets. Further more, the return to small satellite missions was also driven by the technology advancements. Thus, small satellites could be developed as they are not only providing valuable scientific returns, but also allow in the cost effective solution and completely new applications in remote sensing, environmental monitoring, communications, rapid response science and military missions, and technology demonstrations. The spirit of the small satellite world has been encompassed by the slogan "Faster, Better, Smaller and Cheaper". Although various high profile failures of NASA spacecraft have made this phrase less popular [22]. Nevertheless, small satellite projects are characterized by rapid development scales for experimental missions when compared with the conventional space industry, with kick-off to launch schedules ranging from just six to thirty-six months.

There is no universally accepted definition of a “small satellite”. Usually an upper-limit of about 1,000 kilograms is adopted. Below that limit, satellites over 100 kilograms are frequently called “minisatellites” or “enhanced microsatellites”, between 10 and 100 kilograms “Microsatellite” and below 10 kilograms “Nanosatellite”[22]. At the University of Surrey, England and Northern Ireland, satellites having a mass between 500 and 1,000 kilograms are called “small” and between 100 and 500 kilograms “mini”. The European Space Agency (ESA) usually considers 350-700 kilograms satellites “small”, 80-350 kilograms “mini” and 50-80 kilograms “micro”. The cost of developing and manufacturing a typical minisatellite is between US$ 5 million and 20 million, a microsatellite between US$ 2 million and 5 million and a nanosatellite could be below US$ 1 million. In this thesis the generic term “small satellite” is used for spacecraft of less than 500 kilograms.

Many of the space missions will require much more capable spacecraft within smaller sizes. These will include high precision earth observation and space monitoring, satellite inspection, distributed platforms, constellations, satellite docking and miniature interplanetary probes. Attitude control system is a critical subsystem for all types of spacecraft. This subsystem is needed so that for example the optical system covers the programmed ground area at all times. However, the satellite tends to change its orientation due to torque produced by the environment (drag of the residual atmosphere on the solar array, solar radiation pressure, aerodynamic drag, gravity gradient and magnetic field disturbance) or by itself (due to movement of mechanical parts). Thus, the angular orientation has to be actively controlled. The attitude is continuously controlled by a programmed control loop: sensors measure the satellite's attitude; the onboard computer then processes these measurements and generates commands which are carried out by the actuator to ensure correct pointing of their instruments, antennas and solar arrays in the desired direction. The conventional attitude control subsystems are involved with different actuators such as reaction wheels, thrusters and magnetic torquers.

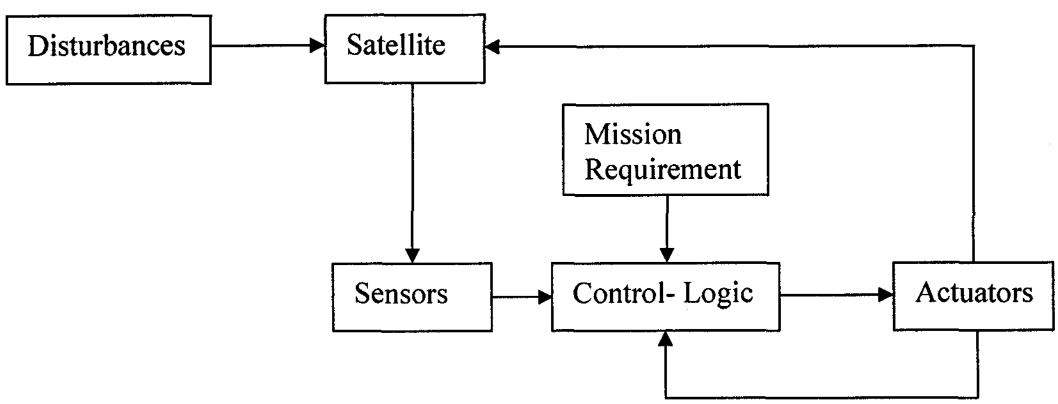

Fig. 1.1: Attitude control task.

1.2 Background

Each spacecraft consists of various scientific instruments selected for a particular mission, supported by basic subsystems for electrical power, trajectory and orientation control, as well as for processing data and communicating with ground station. Electrical power is required to operate the spacecraft instruments and systems. NASA uses both solar energy from arrays of photovoltaic cells and small nuclear generators to power its solar system missions. However, the power generator is typically an array of solar cells, either attached to the spacecraft exterior, or to articulated solar panels. A backup power source is required during the eclipse periods and the peak power demands. This power source typically comprises of a battery made of electrochemical cells, such as nickel cadmium, nickel hydrogen or lithium ion cells. The rechargeable batteries are employed for backup and supplemental power. As with any technology, there are some limitations associated with chemical batteries. The number of charge-discharge cycles is limited and operational temperature range is narrow. To obtain the higher charge status it is difficult and energy storage and power capacity are coupled, and in general this becomes the limiting factor on the satellite lifetime. Therefore, it is clear that alternative energy storage devices are required to be reinvestigated. One alternative to batteries is a high speed flywheel energy storage.

In addition, the rotational speed of the flywheel generates momentum that can also perform the attitude control for spacecraft. By integrating the energy storage and attitude control functions, flywheels could be used as one single energy storage and attitude control device forming a "Combined Energy and Attitude Control System" (CEACS) in space missions.

Low Earth orbiting (LEO) satellites in particular can benefit from this technology because of the large number of charge/discharge cycles they experience. Significant additional benefits are realized at the satellite system level when the momentum storage capability of flywheels is utilized for attitude control by combining what are currently two separate functions into one unit. CEACS benefits in LEO include dramatic improvements in cycle life, weight, efficiency, maintenance, cost and logistics. This combination could lead to different advances on mission design, e.g., mass saving, performance enhancement and reliability increase [16]. According to the NASA Lewis Research Center, it has also been studied the feasibility of flywheel energy storage system for GEO satellite. However, the CEACS is more attractive in the LEO missions than in the GEO missions due to high charge/discharge cycles in the latter missions.

The CEACS consists of a composite flywheel, magnetic bearings, a motor/generator and control elements for the energy and attitude management [3, 7, 16, 18, 19, 20]. A higher rotational speed is expected in order to achieve a higher energy storage capability [11, 13]. In the past this application was limited for large satellites (e.g., ISS). In the case of small satellites, the magnetic bearing causes magnetic interference to the iron parts of the motor-generator [15]. Therefore, the use of ironless motor-generator for this combined system is essential, which will ensure the CEACS flywheels to be controlled efficiently in torque mode [18].

1.3 Problem Statement

Combining the conventional energy and attitude control system is a feasible solution for small satellites to improve the space missions. In this combined energy and attitude control system (CEACS), a double counter rotating flywheel is used to replace the conventional battery for energy storage as well as to control the attitude of earth oriented satellites. In the past years, this system was proposed for large spacecraft, but until today the feasibility of CEACS has not been end-to-end analyzed on small satellites, such as the micro/nanosatellites. The attitude control design can be implemented either based on a speed control mode or a torque control mode. The speed control based architecture has already been developed to demonstrate the satellite attitude performances [18, 19, 20, 21]. But the torque control mode CEACS is not investigated yet. Therefore, it is a great need to fill this technical gap. In addition, an ironless DC motor/generator presents a higher degree of linearity of current to torque conversion and this property supports CEACS in torque mode. Hence, the key issues in this work are to investigate the torque mode based CEACS attitude performances for different small satellites (i.e., Nanosatellite (10 kg), Microsatellite (50 kg) and Enhanced Microsatellite (100 kg)).

1.4 Objective

This CEACS attitude control architecture can be operated either with a pre-filtered attitude feedback or without a pre-filtered attitude feedback. Unlike the previously developed CEACS attitude control architecture with a pre-filtered attitude input [20], a torque mode architecture is developed herein where the pre-filter is not incorporated. Hence, the energy and attitude inputs for the flywheel control architecture are considered in the torque mode in this research. The CEACS mathematical models are developed using the linear control theory, then the models are simulated through Matlab™ for studying the attitude performances. It is important to mention that the satellite attitude performance of the torque mode is of paramount